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Challenges in the development of a slow burning solid rocket booster

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ABSTRACT

Solid rocket motors always find a place in launch vehicles, especially as booster stages on account of their capability to deliver higher thrust needed during the initial phase of flight, most importantly during lift-off phase. The newly developed slow burning solid booster is chosen for the first stage of Reusable Launch Vehicle-Technology Demonstration (RLV-TD) flight programme. This suborbital two stage technology demonstration programme is aimed for a hypersonic re-entry of double delta wing vehicle from an altitude of nearly 65–70 km, where the first stage of vehicle is identified as solid booster. Therefore, the desirable Mach number of the vehicle at motor burnout (30–35 km) is essentially required to be minimum 6 to attain hypersonic re-entry. The contradictory mission requirements like achieving the desired Mach number and limiting the dynamic pressure demanded for design, development and qualification of a new motor with slowest burning solid propellant thereby an action time of nearly 90 s. The challenges faced in the design, development, realization and qualification of the motor through two successful static tests are discussed in this paper. The special attention given for motor case with 2 mm shell thickness, development of a slow burn rate propellant, ignition strategies for the slow burn rate propellant, stable burning of the low burn rate propellant and flow separation related issues for ground test are highlighted.

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1. Introduction

Adoption of Reusable Launch Vehicle (RLV) instead of the current expendable launch vehicle (ELV) is being conceived as the future route to achieve low cost, reliable and on demand access to space. Studies clearly indicate that, two stage to orbit (TSTO) Reusable Launch Vehicle with a wing body configuration of first stage capable of landing horizontally like an aircraft and a second stage capable of landing vertically or horizontally is the most cost effective configuration for fully Reusable Launch Vehicle system. The hypersonic flight experiment (HEX) mission is the first step towards the demonstration of the technology. A solid motor has been selected to propel the two stage RLV-TD vehicle to the desired altitude with a Mach number greater than 6. In order to meet the stringent mission requirements, a new motor has been designed, realized and successfully qualified. The head and middle segments of the solid booster are cylindrical grain ports whereas

the nozzle end segment is ten lobe deep slotted star configuration selected to avoid erosive burning. When the flow velocity over the grain surface is above a threshold value, burn rate increases drastically. For motors with large L/D or lower A_p/A_t ratio, pressure during initial few seconds will be higher. This behavior is called erosive burning.

The nozzle is straight conical convergent-divergent type with secondary injection thrust vector control (SITVC) system. All the nozzle liners including throat insert except SITVC inserts are made of carbon phenolic. SITVC with 3 pintle system is provided at four diametrically opposite locations at 90° apart in divergent section for pitch and yaw control during initial 10 s of motor action during the flight. Then the system will be in back up mode with fin tip control system. To achieve the action time requirement of nearly 90 s with a thrust level of 300 kN, a new slow burning propellant with a burn rate modifier is formulated. This new formulation is scaled-up to 350 kg level and several tests were carried out in the standard ballistic evaluation motor configuration. In the next stage of scale up, 1 t propellant mixing is done. One subscale simulation motor is cast from the mix. The simulation motor is designed to operate at the average pressure of the main motor with the same area ratio. Subscale motor was successfully static tested on 15th March 2007 and performance parameters were as per the predication. Stable burning of propellant at low average pressure (1.47 MPa) has been confirmed.

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Nomenclature

Q- α	total Dynamic pressure times total angle of attack	g_0	Standard acceleration gravity at sea level, 9807.0 mm/s ²
γ	Flight path angle	n	Pressure exponent of ballistic steady burning rate, non-dimensional
H	Augmented heat flux	P_c	Combustion chamber pressure
H_u	Flux due to undisturbed flow	r_b	Burning rate
h	Equivalent protuberance height	Γ	Dimensionless parameter defined in equation
δ	Boundary layer thickness	C_D	Coefficient of discharge
n	Index derived from post static test data of earlier motors with SITVC	R	Gas constant
P_{DL}	Pressure below which a propellant will not burn	M	Molecular weight of the combustion products
a	Pressure co-efficient of ballistic steady burning rate, non-dimensional	c^*	Characteristic velocity
A_b	Area of the burning surface	I_{sp}	Specific impulse
A_t	Area of the nozzle throat	ρ_p	Propellant density
T_c	Combustion chamber temperature		

The qualification of the new motor with SITVC was carried out through two static tests. The first static test (ST-01) of this motor was successfully carried out on 19th November 2008. All the subsystems performed satisfactorily as expected. Motor performance and post test assessment of subsystems confirmed availability of required margin. However revised mission requirement necessitated modification to bring down the dynamic pressure. Web thickness at the star portion of the grain has been reduced and demonstrated in the second static test. The challenges faced in the design, realization and testing of the new motor is presented comprehensively.

2. Mission requirements

The specifications prescribed to meet the mission profile are in Table 1, booster stage shall be configured to boost the Technology Demonstrator (TD) payload to a desired Mach number and altitude.

2.1. RLV-TD mission requirements/specifications

Arriving at the desired thrust profile is a trade-off between the mission requirement and practical possibility from a solid motor. Project (RLV-TD) undertook mission optimization studies considering several ideal thrust-time profiles generated for the solid booster. The most optimum ideal thrust-time profile which meets

Table 1
RLV-Solid Booster performance requirements/specifications.

Sl. No.	Parameter	Specification
1	Mach number at motor burn out	>6
2	Dynamic pressure (kPa)	≤ 34
3	Q- α total (Parad)	≤ 180
4	Flight path angle (γ) at burn out ($^\circ$)	<25

all the mission requirements was derived by the above studies and is indicated in Fig. 1. The above sharp ideal thrust-time profile signifies the grain design to be made in such a way that lower thrust of nearly 160 kN to be achieved after first peak at around 310 kN over a period of 30 s. This requires a drastic change of the burning surface area just after the peak. The performance requirements/specifications for RLV-Solid Booster (RLV-SB) derived/translated in to the propulsion parameters after several rounds of refinements is given in Table 2.

3. RLV-SB motor grain design challenges

The real challenge for the grain design team was to achieve an action time of 90 s within the diameter constraint of 1 meter for the motor and meeting the prescribed Thrust-time profile. The challenge involved can be well appreciated by the comparison of

IDEAL CURVE FOR RLV SOLID BOOSTER

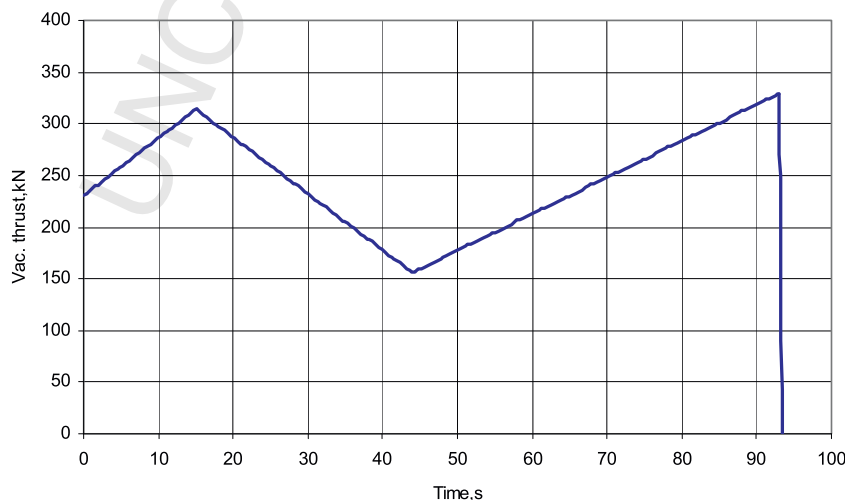


Fig. 1. Ideal Thrust-time profile for RLV solid booster

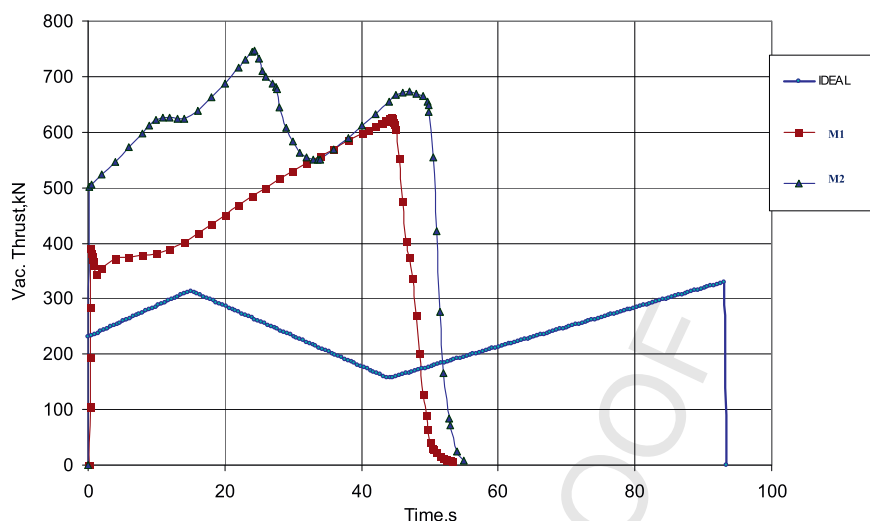


Fig. 2. Comparison of RLV-SB (ideal) T-t profile with existing motors.

Table 2

RLV-SB ballistic performance requirements/specifications.

Sl. no.	Parameter	Specification (preliminary design review)
1	Propellant mass (kg)	$\cong 9200 \pm 50$
2	MEOP (MPa)	< 2.55
3	Action time (s)	$\cong 90 \pm 3$
4	Max. thrust (kN)	$\cong 320$
5	Time at $F_{\max}$ occurs (s)	$\cong 11$
6	Total impulse (MN-s)	$\cong 23.6$

T-t profiles of existing motors with the ideal profile prescribed by the mission as shown in Fig. 2.

RLV-Solid Booster required an action time of nearly 90 s compared to maximum of 50 s for existing motors of same size. More than hundred design options were studied by the grain design team to arrive at the final acceptable configuration.

4. Motor design features

Overall features of the proposed motor as given below:

- New slow burning propellant composition to cater the mission requirement.
- Special grain configuration (combination of cylindrical and star) to meet the thrust time profile.
- Suitable igniter to ignite the slow burning rate propellant with required margin. In order to have a smooth transient at the beginning, a new igniter is design and qualified with higher web thickness and longer grain length. This ensures the sufficient overlap of igniter functioning with motor pressure rise to avoid any abnormalities or ignition delay of slow burning propellant or propellant port surfaces. The inducted igniter in this booster is capable of delivering higher heat flux, sufficiently high mass flow rate and longer burn time compared to the existing igniter for 1 m class of solid booster.
- Motor case shell thickness optimized to 2 mm to trim the overall inert weight of the vehicle.
- Modified segment joint with shaft and face seal to enhance the reliability of joint interfaces.
- Three pintle SITVC at four locations for generating maximum 11 kN side thrust requirement.
- Carbon phenolic throat insert instead of graphite and single CP divergent liner.
- Improved motor/nozzle interface by providing wiper face seal.

4.1. Motor case and nozzle hardware design analysis

High strength steel material is chosen for motor case and nozzle hardware considering the pedigree.

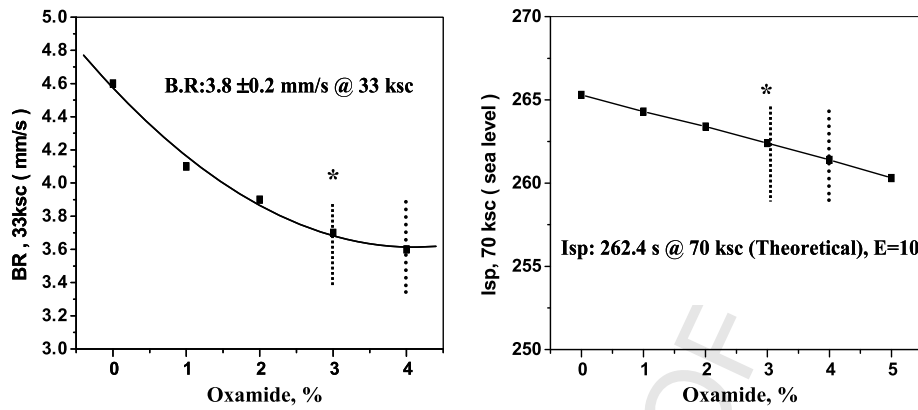
- Cylindrical shell thickness is arrived at based on membrane shell theory. The minimum thickness based on yield is 1.90 mm.
- Required thickness selected is $2^{+0.2/0}$ mm considering scaling in heat treatment.
- The joint has been redesigned corresponding to a load of 3.23 MPa internal pressure against 6.37 MPa in the present motors.

4.2. Motor propellant system

Heterogeneous propellants are mixtures of crystalline oxidizer particles blended within a polymeric fuel matrix. The commonly used oxidizers such as AP and AN produce high oxygen concentration on thermal decomposition. The fuel used is the hydrocarbon-based polymers such as HTPB, CTPB, and PBAN. Typically high concentrations of oxidizers are used to give high specific impulse. Aluminum particles are usually added to further increase the specific impulse. A conventional composites propellant contains 65 to 70% AP, 5–22% Al, 8–14% binder HTPB/PBAN. The perchlorates used for propellants and explosives are ammonium perchlorate (AP: NH_4ClO_4) potassium perchlorates (KP: KClO_4), and nitronium perchlorate (NP: NO_2ClO_4). AP is the major crystalline oxidizer used for composite rocket propellants. The oxygen molecules produced by the decomposition act as an oxidizer when AP particles are mixed with polymeric fuel components. In addition to the various techniques used to control the burn rate of the propellant, additives play major roles to bring down the burn rate and plume temperature [1–4].

In general, burning rates depend on:

- Nature of energetic material (basic ingredients and their mixture ratio);
- Details of chemical composition (catalysts, modifiers, additives, etc. usually present in small or fractional percentages);
- Physical effects (particle size distribution, presence of wires or staples, etc.);
- Details of manufacturing process and other miscellaneous factors;



* Lines indicate the detailed analysis was carried out at 3% and 4 % of oxamide to meet performance

Fig. 3. Effect of burn rate and specific impulse with moderator.

- o Operating conditions (pressure, initial temperature, natural and/or external radiation, heat losses, gas flow parallel to the burning surface, acceleration, etc.);
- o Mode of operation (steady vs. unsteady).

The most effective coolants are those with decomposition temperatures near that of the surface temperature of the burning propellant. Compounds with low melting or decomposition temperatures show little effect. Although the coolants produce changes in the mass burning rate with pressure, they do not significantly increase of P_{DL} [5,6].

A burn rate suppressant is an additive that has the opposite effect to that of a catalyst, it is used to decrease the burn rate. For AP based propellants, oxamide (NH_2CO_2) is particularly effective in reducing burn rate, without sacrificing performance. Other potential burn rate suppressants include calcium carbonate, calcium phosphate, ammonium chloride, and ammonium sulphate and typically 1–3% oxamide is used which results in 20–30% burn rate reductions with very little affect on I_{sp} . The oxamide reduces the decomposition of ammonium perchlorate, therefore the forward reaction suppressed and hence lowered the flame temperature and burn rate. The effect of additives whose decomposition temperatures are higher than the surface temperature of the matrix propellant, ammonium perchlorate (AP) was added to control nitrocellulose (NC)-based propellant. AP is known to have a higher surface temperature than NC double base propellants. AP (even though it is an oxidizer) does not appreciably affect the ignition limit of the NC double base propellant.

To achieve the mission requirements of 90 s action time, a slow burning propellant was developed using oxamide as a burn rate modifier. Large numbers of formulation trials with varying content were carried out to fix its percentage. Scale up studies carried out with the selected composition. Fig. 3 comprises the variations of burn rate and I_{sp} with modifier. Both I_{sp} and burn rate shows decreasing trend with increasing modifier. Detailed characterization of final formulation was carried out, repeatability mechanical and interface properties with varying batches of ammonium perchlorate and HTPB were examined. Propellant cure cycle is optimized. The achieved burn rate of the propellant is 3.8 mm/s at 3.43 MPa and burn rate law is $r = 0.14P_c^{0.3}$.

4.3. Motor grain configuration

High velocity gas flowing parallel to the burning surface can seriously increase the local burning rate (causing the so-called erosive burning phenomenon) due to increased heat transfer from the adjacent turbulent boundary layer, especially in the aft-end portion

of the motor cavity. Motor acceleration more than 10g, whether longitudinal, lateral or due to spinning motion, directed into the burning surface and within an angle of 60 to 90° with respect to it, perceivably increases the burning rates. Other peculiar ballistic effects, due to the details of manufacturing process may be important for the motor operations but are sensibly dependent on the actual configuration. Several grain configurations are worked out to finalize the grain configuration of the segments to avoid the unsteady flow and erosive burning phenomenon. The burning surface of solid energetic materials regresses in a direction perpendicular to itself. In other words, solid propellants are considered to burn by parallel layers and the grain “tends to retain its original configuration until the web has burned through” [6,7]. This is the philosophy consider for radial burning propellant.

The initial grain geometry of a solid propellant strictly depends on the propulsive mission. The uniform pressure and burning rate throughout the combustion chamber of a solid propellant rocket motor filled with a perfect gas burned mixture. All properties are considered constant. Transient mass conservation requires:

$$m_g = m_d + \frac{d(\rho_c V_c)}{dt}$$

where the mass production of gas due to combustion is given by

$$m_g = \rho_p A_b r_b$$

the mass flow rate exiting the nozzle is

$$m_d = \frac{p_c A_t}{C_D} = p_c A_t c^*$$

and the mass accumulation rate in the combustion chamber is

$$\frac{d(\rho_c V_c)}{dt} = \rho_c \frac{dV_c}{dt} + V_c \frac{d\rho_c}{dt} = \rho_c A_c r_b + V_c \frac{1}{\Gamma^2 c^{*2}} \frac{dp_c}{dt}$$

where

$$c^* = \frac{1}{\Gamma} \sqrt{\frac{R}{M} T_c}$$

is the characteristic velocity.

By substitution in the mass conservation equation, one finds the transient equation of the internal ballistics

$$\frac{dp_c}{dt} = \frac{\Gamma^2 c^{*2}}{V_c} (\rho_p - \rho_c) A_b r_b - \frac{\Gamma^2 c^*}{V_c} A_t p_c$$

Under steady operations, one obtains the equilibrium pressure of the rocket motor combustion chamber

$$p_c = \left(\frac{A_b}{A_t} a_b \rho_p c^* \right)^{\frac{1}{1-n}}$$

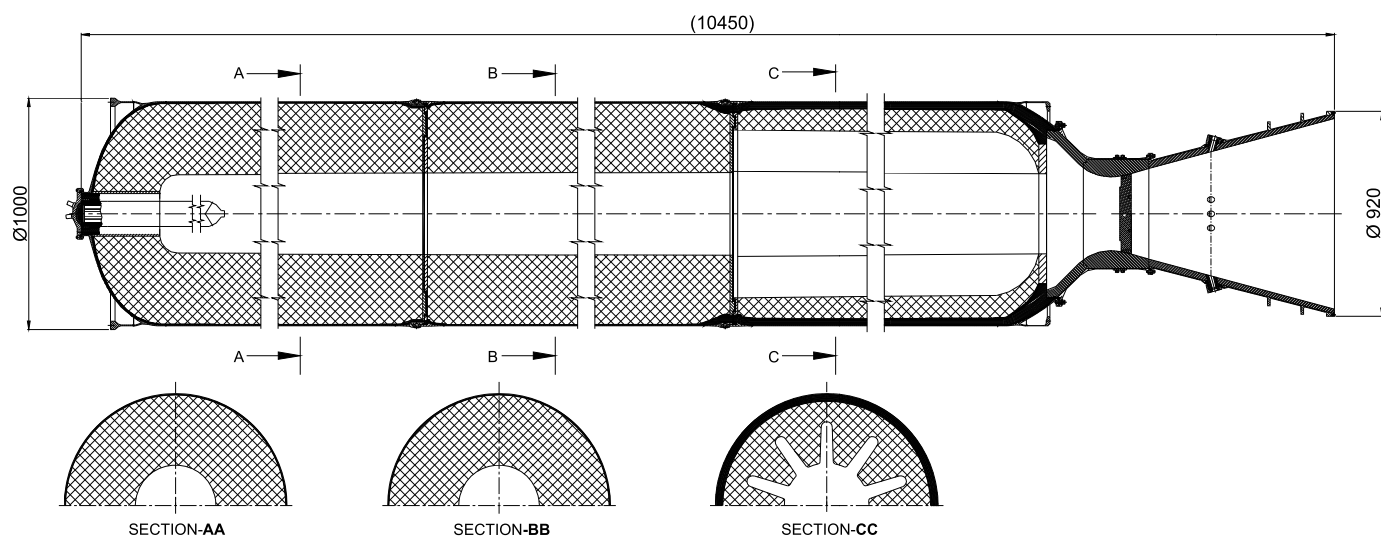


Fig. 4. Motor grain configuration.

where the steady burning rate has been taken as

$$\bar{r}_b = a_b p_c^n, \quad \text{where } \rho_p \gg \rho_c$$

Design and operation of solid rocket motors strongly depend on the combustion features of the propellant (burning rate, burning surface and grain geometry) and their evolution in time. Internal ballistics is the applied science devoted to these problems.

The final motor grain configuration arrived is as follows, cylindrical for both head end, mid segments and 10 lobe deep-star for nozzle end segment to achieve the required thrust-time profile as shown in Fig. 4. The derived trust-time profile is the combination of progressive and regressive type (deep M-type curve).

4.4. Insulation and inhibition design

Well proven existing formulation of nitrile rubber is chosen as the material for motor case insulation. The insulation design of the motor is carried out by thermal response study, based on time to exposure and exposure duration. Thermal erosion is computed using char ablation model. The various post test data from existing motors are analyzed and used for predicting the mechanical erosion. The erosion rate of carbon phenolic throat varies from 0.066 to 0.076 mm/s.

The design thickness is arrived at based on expected erosion at each location. Design thickness (D_T) = $1.35 \times \text{Erosion} + \text{Virgin thickness}$ required to limit back wall temperature at 373 K.

4.5. Slow burning propellant composition development and scale-up

The new formulation was scaled up to 350 kg level and several tests were carried out at 40 kg propellant level. Further to scale up the new slow burning propellant, a simulation motor with 1 t of propellant has been tested simulating the main motor average pressure and area ratio. The subscale motor test conducted to evaluate the vacuum specific impulse of the formulation. The measured vacuum specific impulse was 260 s against the CFD analysis of 259.80 s. The subscale motor test ruled out the possibilities of chuffing and combustion instabilities at lower chamber pressures. The burning rate of the standard small-scale motors with defined operating conditions is calculated by using typical methods based on web over time and mass balance. While the burning rate at motor level (under the same defined operating conditions) is extracted by comparing theoretical and experimental pressure-time traces and imposing that a minimum error occurs during tail-off phase.

The steady burn rate pressure dependence of ammonium perchlorate (AP)-based composite propellants is conveniently fitted by the Vieille-St. Robert law [8]. Experiments were carried out to evaluate the pair of ballistic parameters 'a' and 'n'. Several tests were performed using multiple small test motor (STM) with different throat diameters. Each test produces a recorded pressure-time profile, yielding effective burn rate and pressure by an appropriate data-reduction procedure. The propellant ballistic properties are defined by fitting at least three different pressure levels and the corresponding burn-rate values.

Six STM grains are cast, four STM are tested at around motor average pressure to assess the burn rate dispersion. One STM is tested at minimum pressure and sixth STM is tested at motor Maximum Expected Operating Pressure (MEOP). Based on this test result, burn rate law is established for the operating pressure range of the motor. The final propellant formulation after scaled up to one ton mix used for this motor includes 84% ammonium perchlorate (AP), 11.8% hydroxy-terminated polybutadiene (HTPB), 4% oxamide, 0.1% Ambilink, remaining with plasticizer [3,4].

5. Nozzle design

Nozzle plays a pivotal role in solid motors. The design of nozzle internal contour usually requires to consider the demand of mass flow, the thrust, the thermodynamics characteristics of combustion gas, environmental characteristic (to assure the least loss), and smallest size and weight of the nozzle and so on. The nozzle is straight conical convergent-divergent type with SITVC. Since the motor is operated at lower altitude, the preferred area ratio of 6 to 7 is finalized. The variation of Mach number along the area ratio of nozzle contour from the convergent to divergent exit is plotted in Fig. 5. Mach number varies from 0.2 at convergent fore end to 3.5 at divergent aft end as plotted in Fig. 5. When ambient pressure considerably greater than nozzle exit pressure can induce boundary-layer separation, oblique shocks and flow separation in the case of over expanded nozzles. In such case the flow can separate during the tail-off region and produce more thrust. The governing equation of Sutherland's empirical relations by the help of CFD tool is used for flow separation study. The study clearly reveals that, flow separation starts at 10 s and separation ends at 20 s. The minimum nozzle area is called the throat area. The maximum gas flow per unit area occurs at the throat where there is a unique gas pressure ratio which is only a function of the ratio of

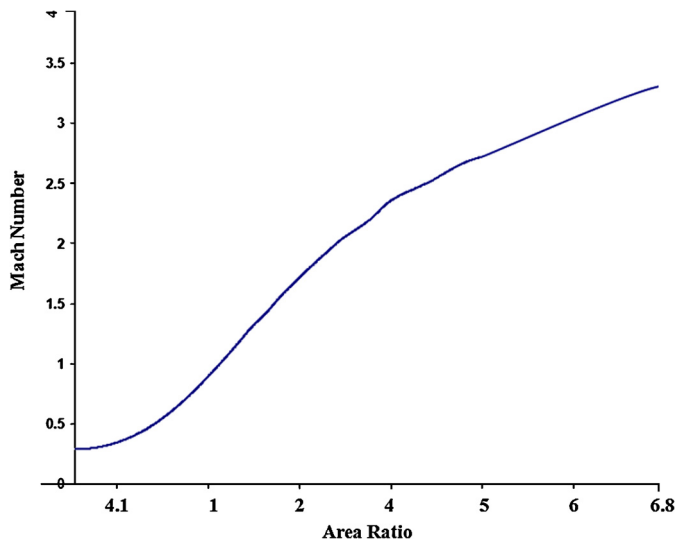


Fig. 5. Variation of Mach number with area ratio of nozzle.

specific heats. At the point of critical pressure the Mach number is sonic.

All the nozzle liners except SITVC inserts are made of carbon phenolic. SITVC is provided at four diametrically opposite locations in divergent for pitch and yaw control during initial 15 s of motor operation during flight. Then the system is in back up mode with fin tip control system.

Erosion in convergent and throat liner is governed by two distinctive mechanisms given below:

(i) Erosion due to convergent heat transfer: thermal erosion due to convective heat flux predicted by nonlinear charring ablative model.

(ii) Erosion due to impingement of alumina particles: Chiba's co-relation is used for the evaluation of impingement erosion in the convergent and throat insert. It correlates impingement erosion to velocity of gases and angle of impingement [8].

5.1. Convergent sub-assembly

The convergent sub-assembly consists of a conical convergent and cylindrical throat housing made of high strength steel material. The convergent liner is rosette layup and cured in autoclave.

5.2. Divergent sub-assembly

The divergent sub-assembly consists of high strength steel material back-up hardware and 0° tape wound carbon phenolic liner which is cured in autoclave.

5.3. SITVC ports

The SITVC injection plane is at 45% of throat to exit length with 15° injection angle. There are four SITVC ports (3-pintle) at 90° apart as shown in Fig. 5. The SITVC system is chosen due to simplicity and less inert weight over flex nozzle system. Secondly, control system is only required for 10 s initially. As a result, the left out liquid injectant will be flushed out as given in Fig. 6. Aqueous strontium perchlorate is selected as injectant. Strontium perchlorate is selected as a candidate material for SITVC injectant because of non-corrosive, non-hazardous and relatively higher density compared to other injectants [9]. The port is lined with random fibred oriented silica phenolic back-up liner. The port bore is provided with another silica phenolic liner with random fiber. For the same flow rate, three pintle injection valves have high injection efficiency compared to the single pintle valves. The SITVC system will

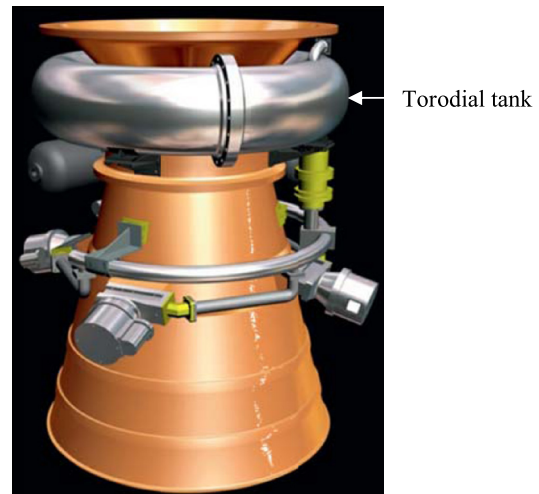


Fig. 6. Nozzle with SITVC mounting interfaces.

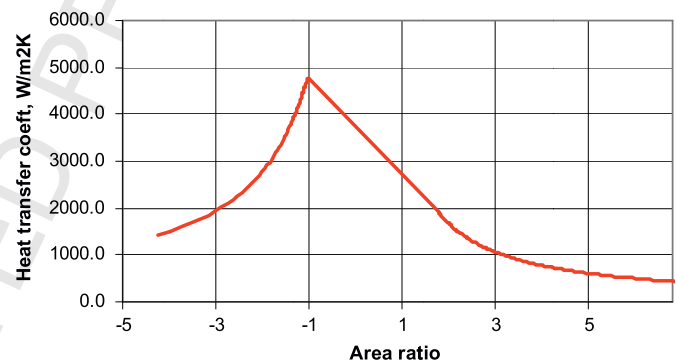


Fig. 7. Heat flux along nozzle contour.

employ electro-mechanical injection valve instead of electro hydraulic system for simplicity and to avoid messy hydraulic system.

5.4. RLV-SB nozzle system

Heat flux is computed using Bartz equation. Computations are based on the two dimensional charring ablation model. Design thickness = $1.25 \times \text{char depth} + 1.1 \times \text{virgin thickness}$ required to maintain interface temperature below 100 °C at burnout [10–12]. Erosion due to convective heat transfer and particle impingement was considered for deriving the nozzle liner thickness. The variation of heat transfer coefficient with nozzle area ratio is given in Fig. 7. Thermo-physical properties of the ablative liner obtained from test data are corrected for appropriate fiber orientation and taken as input data for the design and analysis. The heat flux augmentation due to SITVC injection depends upon the equivalent protuberance height and boundary layer thickness. The equivalent protuberance height of the boundary layer thickness is taken as the parameter for heat flux augmentation estimation. The correlation used is $H/H_u = [h/\delta]^n$ [11], H is the augmented heat flux, H_u is the flux due to undisturbed flow, h is equivalent protuberance height, δ is the boundary layer thickness and n is index derived from post static test data of earlier motors which used SITVC system. To build up initial pressure for smooth combustion of motor, a closure is provided at the divergent side of the convergent sub-assembly location at 50 mm away from throat and 80 mm inside the flange face of the convergent hardware exit. It is designated for the burst pressure of 9 ± 3 ksc internal pressure.



Fig. 8. RLV-Solid Booster static test.

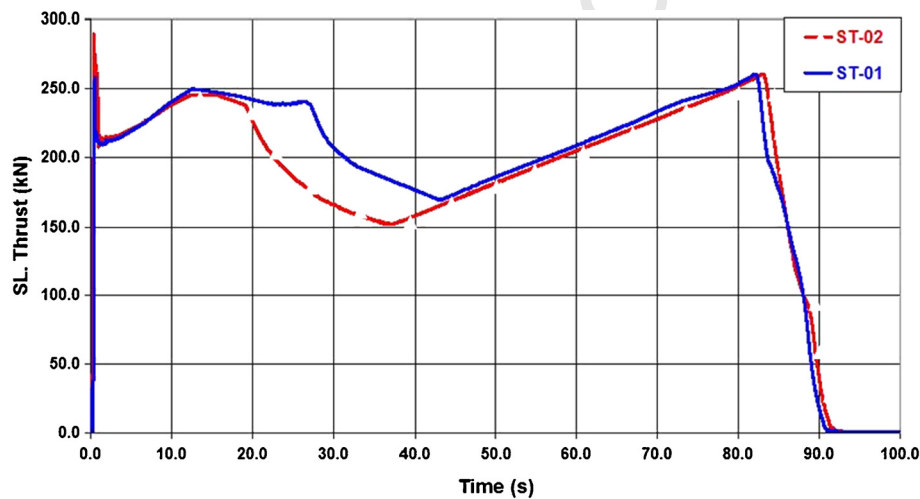


Fig. 9. Comparison of sea level thrust data of ST-01 vs. ST-02.

6. Static tests

The qualification of the new motor with SITVC was carried out through two static tests. The first static test (ST-01) of this motor was successfully carried out on 19th November 2008. All the subsystems performed satisfactorily as expected. Based on the detailed post test assessment and the revised mission requirement certain modifications are incorporated for the second static test motor. The propellant web thickness of nozzle end segment is reduced by 30 mm to modify the thrust time profile so as to limit the peak dynamic pressure at transonic regime of the vehicle. The igniter design is modified to have sufficient overlap of igniter functioning with the motor pressure rise. The motor was modified and the second static test (ST-02) was successfully conducted on 3rd August 2011. Panoramic view of static test is given in Fig. 9.

7. Performance of the static test motors

The nozzle throat diameter conceived for performance analysis is 335 mm with average STM burning rate of 3.1 to 3.2 mm/s @ 33 ksc. The realized propellant mass is nearly 9400 kg and 9360 kg for ST-01 and ST-02 respectively. Pressure index of 0.251 is assumed for grain design analysis. The achieved burn time and action time from both the test are of the order of 80 s and 90 s respectively. The observed maximum chamber pressure is limited to 2.4 MPa. The generic measured maximum sea level thrust is less than 260 kN and vacuum I_{sp} is 259 s. Comparison of performance plots of first static test and second static test is given below.

From the thrust time plots from Fig. 8, it is evident that the thrust level of ST-02 has come down, due to addition of 30 mm insulation thickness on nozzle end segment. The integral pressure up to 40 s of ST-02 is less than the ST-01. This is the most favorable design as per mission is concerned, which essentially control the peak dynamic pressure of the vehicle during transonic regime.

8. Post test inspection results and overview

The O-rings were intact and overall health of post fired motor and nozzle was found in good condition. No soot or scorch mark noted at the segment joint in second static test. The foot prints of star and lobe pattern is clearly observed in nozzle end segment as shown in Fig. 10a. Observed erosion in convergent and throat is very low. Specifically, the erosion at aft of throat is either zero or negative in all the generators. The health of liner at 3 pintle SITVC port locations is shown in Fig. 10b. Virgin remaining in convergent, throat and divergent gives adequate thermal margins and hence the convergent and divergent sub-assembly is thermally adequate and doesn't call for any design modification. Throat erosion rates are 0.02 mm/s and 0.018 mm/s for ST-01 and ST-02 respectively. Interface of the motor was normal and higher erosion was seen near to SITVC location. Erosion profile of insulation on head end segment (HES), mid segment (MS) and nozzle end segment (NES) are different, however the positive factor of safety (FOS) 1.38, 1.28 and 1.18 is available for HES, MS and NES respectively.

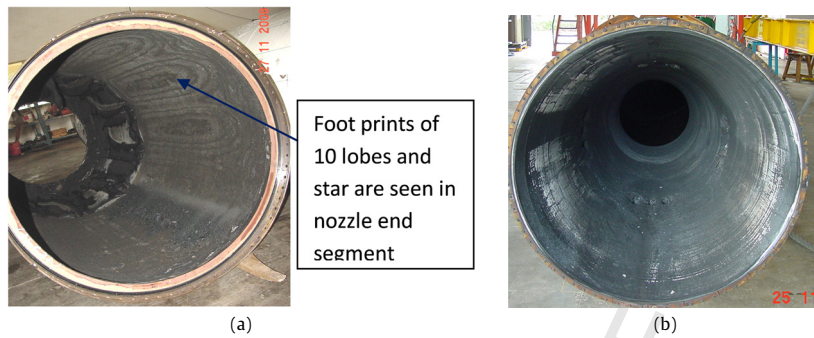


Fig. 10. (a) Post fired motor segments, (b) post fired nozzle.

9. Conclusions

With the completion of two successful static tests, the new motor has been qualified to the stringent mission requirement. No features from the existing 1.0 m diameter motors could be directly adopted because of specific and contradictory mission requirements like 90 s action time, low thrust requirements for limiting the dynamic pressure at transonic regime. Propellant composition, hardware with 2 mm shell thickness for 1.0 m class of motors and ignition system are totally new. After the first static test, grain web thickness in the star portion has been reduced in the second static test to reduce the dynamic pressure. Motor performance was satisfactory and post test measurements inferred availability of sufficient margin in all the subsystems.

Conflict of interest statement

None declared.

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